RESEARCH MEMORANDUM

APPLICATION OF NON-AFTERBURNING TURBOJETS TO SUPERSONIC FLIGHT

By Richard S. Cesaro and Curtis L. Walker

NACA Headquarters
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INTRODUCTION

The range of an airplane can be shown to be proportional to the product of lift-drag ratio and over-all efficiency of the engine. Lift-drag ratios of aircraft with supersonic capability are considerably less than those of aircraft with only subsonic capability. Over-all engine efficiency of afterburning engines used to attain supersonic flight in present aircraft is less than the efficiency of their non-afterburning versions. The purpose of this paper is to consider the feasibility of attaining, through the use of non-afterburning engines, essentially the same range with an all-supersonic mission as the range currently obtained with a mission incorporating subsonic cruise and supersonic dash.

DISCUSSION

In discussing airplane performance, it is convenient to refer to the familiar Breguet range equation.

$$\text{Range} = h\,\eta_e\,L/D\,\log_e \frac{1}{1-W_f/W_g} \qquad (1)$$

The first term, h, is the heat of combustion of the fuel expressed as the number of miles for which one pound of thrust is produced by burning one pound of fuel; thus, if all the chemical energy in one pound of a typical JP-4 turbojet fuel were converted into thrust, it would produce one pound of thrust for approximately 2400 nautical miles. Since an engine does not have an efficiency of 100 percent, the distance over which this pound of thrust is available is considerably less than the ideal value. The next term, η_e, is over-all engine efficiency, that is, the ratio of work done on the airplane (thrust times distance flown) to the energy of the fuel consumed in flying the distance. For current JP-4 (jet fuel), the relationship between engine efficiency, η_e, and specific fuel consumption, SFC, is η_e = (airplane speed in knots) /2400 x SFC) or above 35,000 feet = (.24) (Mach No.)/SFC, where SFC is in terms of pounds of fuel per hour per pound of thrust. The value, then, of h η_e determines the airplane flight distance over which one pound of fuel will produce one pound of thrust. The number of pounds of airplane weight that this one pound of thrust will support in the air is equal to the lift-drag ratio, L/D, of the airplane, and is therefore introduced as the next term in the equation.

Finally, the percent of gross weight that is fuel, when multiplied as indicated in equation (1) results in the range of the aircraft. The percent of gross weight that is fuel is dependent on the percent of gross weight that is required for airframe, propulsion system, fixed equipment, and military load.

A detailed discussion of the interrelation of these factors is given in Reference 1. The significant values for these terms, Table 1, for a series of bomber aircraft indicate the trends in these variables over the past 15 years. These data are representative of early models of the aircraft listed. The radii listed are computed from Equation (1) assuming that the cruise fuel is 90% of the total fuel for the reciprocating engine bombers and 80% for the turbo-jet bombers. It can be seen that the ratio of structural plus fixed equipment weight to gross weight ratio has been reduced about 50 percent over the past 15 years. Pay load or military load has decreased about 60 percent attended by as much as an 8-fold increase in gross weight. Lift-drag ratio has also increased about 50 percent. The decrease in structural plus fixed equipment weight and the decrease in pay load ratio has permitted a 5-fold increase in fuel to gross weight ratio. The final effect on range, as a result of the marked advances made by the aircraft industry in the areas noted above, is an increase in range of the B-52 type airplane over the B-24 type aircraft of approximately 7-fold, or an increase in combat radius from 440 nautical miles for the B-24 to better than 3,000 nautical miles for the B-52.

The data in Table 1 shows that the supersonic performance over the target required for the B-58 airplane results in a marked reduction in range. This reduction is due to the reduction in lift-drag ratio in the subsonic flight phase as well as in the supersonic region. When compared with the all subsonic machine the L/D at subsonic flight has been reduced between 30 and 50 percent, and the L/D at the supersonic flight speed by approximately 65 to 75 percent.

Attention is now directed at the product of efficiency and L/D in the range equation to determine the extent to which the product could be increased for supersonic flight. The thermodynamic cycle characteristics of the turbojet engine clearly indicate that large gains in over-all efficiency can be obtained by the use of a non-afterburning turbojet engine (see figure 1 a). At Mach 2, for example, the efficiency at a turbine inlet of 1550° F is approximately twice that obtained with an afterburning engine. Although the use of a non-afterburning turbojet engine for supersonic flight has been considered in the past, two factors were primarily responsible for it not being adopted: large frontal areas and high specific engine weights. The thrust per pound of air at turbine inlet temperatures of 1550° F at Mach 2 is approximately 1/3 that of an afterburning system as seen in figure 1 b. Therefore, for this condition the free-stream-tube-capture area for the non-afterburning engine system will be from 2-1/2 to 3 times that for an afterburning engine producing the same thrust. The relationship of engine areas to free-stream-tube-capture area for various flight Mach numbers is presented in figure 2. Two recent developments have made it appear that the non-afterburning turbojet engine should be considered for supersonic flight. First, the design trends in the engine industry toward non-afterburning engines with specific engine weights at sea-level static on the order of 0.10 to 0.25 lbs. per pound of thrust. Second, the possibility that the larger free-stream-tube-capture area can be tolerated without undue penalty in the airplane trimmed lift-drag ratio at supersonic speeds.

In determining the effects of the decreased specific engine weights, certain aspects of engine and airframe matching will be considered. The problem of matching requires an examination of the variables that interrelate the engine and airframe and ultimately determine the range, altitude and Mach number of a given system. The familiar Breguet equation illustrates to a first approximation the key engine and airframe variables, their relationship to one another, and how they combine to finally determine the range of the airplane. For the engine, the specific engine weight and over-all efficiency are important and in the airframe, the trimmed L/D and the airplane weight distribution are important.

In considering the relationship of engine and airframe variables that determine the altitude and Mach number of the system, the following equation, expressing the conditions that must be satisfied in level flight, shows the importance of specific engine weight, L/D of the airplane, and the percent of gross weight that is engine:

$$W_e/F = L/D \cdot W_e/W_g \qquad (2)$$

The above equation states simply that the gross weight divided by the thrust output is equal to the airplane lift divided by the airplane drag. (L/D).

In determining the percent of the gross weight of airplane that must be alloted to the power plant, we will begin by examining the relationship between target altitude and take-off wing loading. This relationship is largely determined by the coefficient of lift (C_L) value at a given Mach number for maximum lift-drag ratio and the percentage of take-off gross weight that is target weight. Figure (3) presents the target altitude as a function of the take-off or begin cruise wing loading for representatave values of W_T/W_{BC}. Once a target altitude has been selected, the take-off or begin cruise wing loading is established.

We now include the engine requirement for the altitude performance desired. To do this, we will next refer to the relationship between target altitude and thrust to gross weight ratio for a turbojet engine. Simplifying equation (2):

$$W_g/F = L/D \qquad (3)$$

Corrected to sea level static conditions:

$$\left(W_T/W_{BC}\right)\left(F_{SLS}/F_T\right)\left(W_g/F\right)\ SLS = L/D \qquad (3\ a)$$

(symbols are defined in Appendix)

Figure 4 shows target altitude versus take-off thrust (which is assumed to be the same as the sea-level static thrust) to gross-weight ratio as derived from the above equation for a non-afterburning engine. A representative L/D value at Mach 2 (which includes any penalty for the larger frontal area of the non-afterburning engine) and representative values for target weight as a percentage of take-off gross weight have been selected. The figure illustrates that for target altitudes from 55,000 to 70,000 feet, the thrust to gross weight ratio at sea level static is of the order of .5 to .9 respectively.

So far, we have considered in a gross sense the general relationship of airframe variables that determine target altitude and take-off or begin cruise wing loading and the thrust to gross weight ratio required by the engine for a given target altitude as determined in equation (3). Now it is necessary to evaluate the compromises that can be made between specific engine weight, percentage of gross weight that is power plant, and percentage of gross weight that is fuel in order to derive a realistic compromise among altitude, Mach number, and range. To do this we next examine the relationship of engine weight, airplane weight and thrust for various target altitudes, considering the non-afterburning turbojet. To illustrate the problem, a flight Mach number of 2 has again been selected for a representative bomber configuration. Figure (5) presents the ratio of engine weight to airplane gross weight versus the ratio of engine installed weight to thrust at take-off required for target altitudes of 55, 60, 65, and 70,000 feet. Installed specific engine weight is defined as the dry specific engine weight times 1.25. Let us now examine the interplay that exists among these variables, and so obtain an appreciation for the range of specific engine weights that can make the non-afterburning turbojet of interest for supersonic flight.

Let us begin discussion of Figure (5) by considering current state of the art in terms of the bomber configuration or long-range interceptor where the installed engine weight to airplane gross weight at sea level static is of the order of 10 percent. At a value of engine weight to airplane gross weight of 10 percent, horizontal intercepts at various target altitudes shown defines a range of installed specific engine weight from approximately .10 to .25 lbs. per lb. of thrust. However, if we are willing to accept a higher value of engine weight to airplane gross weight ratio at sea level static take-off, let us say, for example, 15 percent, we see that non-afterburning engines with installed specific engine weights on the order of .25 to .30 can be considered for all supersonic non-afterburning turbojet application. We have not as yet, however, obtained a clear picture of what this trade in engine weight at the expense of fuel weight means from a range loss standpoint. To obtain this appreciation, figure (6) presents the effect of fuel weight reduction on range. This figure presents the reduction in range (percent) versus reduction in fuel weight (percent of gross weight) for various values of fuel to gross weight ratio available for cruise. It is pertinent to mention here that for bomber type configurations or long range interceptors, the fuel to gross weight ratio available is on the order of 60 percent, whereas, for the interceptor type aircraft current values are on the order of 25 to 30 percent.

A range of fuel to gross weight ratios are indicated because high altitude operation may require structure weight increases. This increase in structure weight, of necessity, would have to come out of fuel weight. Considering an engine weight to airplane gross weight ratio on the order of 15 percent, instead of current practice of around 10 percent for bomber type configurations, and applying that change in power plant weight increase to a reduction in fuel weight, a total range reduction on the order of 12 percent results.

The importance of the non-afterburning system becomes clear when a range comparison is made between the non-afterburning system and the afterburning system, figure 7. In the figure no penalty in the trimmed-lift-drag ratio is assumed for the non-afterburning system because of the greater free stream tube area. The supersonic L/D is assumed to be .42 of the subsonic L/D.

The figure shows, for a 2-point operating machine, the trade that can be made between dash radius and total radius for the two systems. As the dash radius is increased, (dash at Mach 2), the total radius is decreased. This figure illustrates the compromises that can be made between total radius and dash radius. It can be seen that for the assumptions made, a system that employs subsonic cruise for 86% of the mission radius and supersonic dash with afterburner for 14%, the relative range is the same as that for the entire mission performed at supersonic speeds, without the afterburner. There is an attendant gain in cruise altitude with the non-afterburning system over that obtained with the 2-point operating system (subsonic cruise-supersonic dash) which is directly attributed to the higher cruise Mach number. The figure also indicates that for all supersonic performance, using the afterburner, a considerable reduction in total radius results over that possible with a non-afterburning system.

In preparing figure 7, as mentioned, the supersonic trimmed lift-drag ratios for the afterburning and non-afterburning systems were assumed to be the same. Much research is being conducted on methods to decrease the effect of the engine nacelle or inlets on the lift-drag ratio of the airplane, and encouraging progress is being made. Nevertheless, it is difficult to estimate without a detailed design study the extent to which the assumption can be approached. Any reduction in the lift-drag ratio would result in a proportional reduction in the curve for the all supersonic non-afterburning system. For instance, a 10 percent reduction in L/D for the non-afterburner supersonic system would mean a total supersonic range equal to that for a 75% subsonic, 25% supersonic afterburner dash system.

CONCLUDING REMARKS

Advancements in "state-of-the-art" engine design as related to engine specific weight have permitted consideration of non-afterburning engines for supersonic flight with little or no change in the percentage of engine airplane gross weight ratio over that employed with current airplanes. This factor indicates that an all supersonic mission with ranges equivalent to the current subsonic cruise-supersonic dash missions utilizing non-afterburning engines has feasibility if the larger free-tube-stream-capture areas associated with non-afterburning engines can be utilized without undue penalty in the trimmed lift-drag ratio of the airplane.

Headquarters
National Advisory Committee for Aeronautics
Washington, D. C., November 18, 1955

APPENDIX

SYMBOLS

C_L, coefficient of lift

F, net thrust

F_T, net thrust at target

h, heat of combustion

L/D, airplane trimmed lift-drag ratio

M, flight Mach number

S, wing area

SLS, sea level static

W_{af}, airframe weight

W_{BC}, airplane weight at begin cruise

W_e, engine installed weight

W_f, fuel weight

W_g, airplane gross weight

W_{pl}, payload

W_T, airplane target weight

η_e, overall engine efficiency

REFERENCE

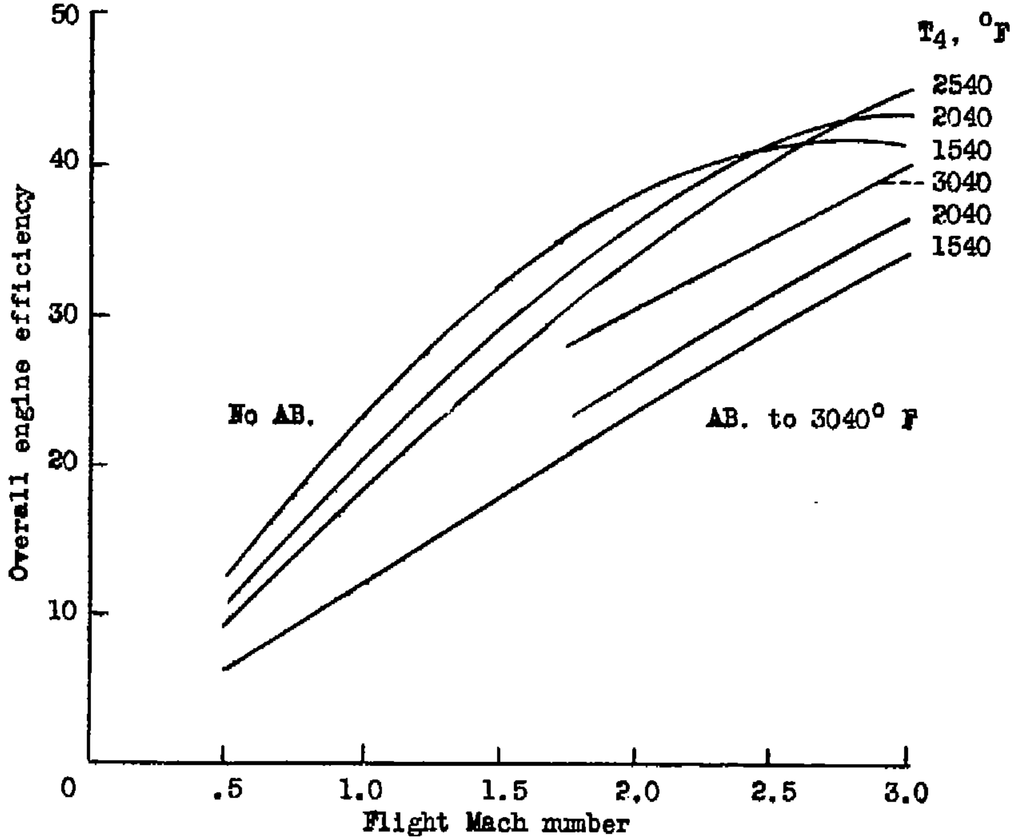
Rothrock, Addison M.: Turbojet Propulsion-System Research and the Resulting Effects on Airplane Performance. NACA RM 54H23, 1955.

TABLE I - PERFORMANCE DESIGN TREND

Airplane	Airplane gross weight	Ratios to airplane gross weight: Engine weight W_e/W_g	Fuel weight W_f/W_g	Airframe weight W_{af}/W_g	Payload weight W_{pl}/W_g	Wing loading W_g/S	Flight Mach number M	Trimmed lift-drag ratio L/D	Over-all engine efficiency N_e, %	Radius n.mi.(1)
B-24	56,000	.179	.123	.443	.255	38.	0.3	13.0	24	440
B-29	105,00	.193	.242	.432	.133	81.4	0.47	17.0	28	1400
B-50	164,500	.171	.320	.275	.234	100.5	0.5	16.6	28	1890
B-36	370,000	.105	.502	.335	.058	77.5	0.6	19.4	28	3920
B-47	200,000	.109	.525	.381	.085	133.2	0.78	17.7	18	2080
B-52	450,000	.097	.592	.212	.099	112.5	0.73	20.0	20	3100
B-58(2)	147,000	.104	.620	.226	.05	100.0	0.90 2.0	12.0 5.0	20 20	1970 820

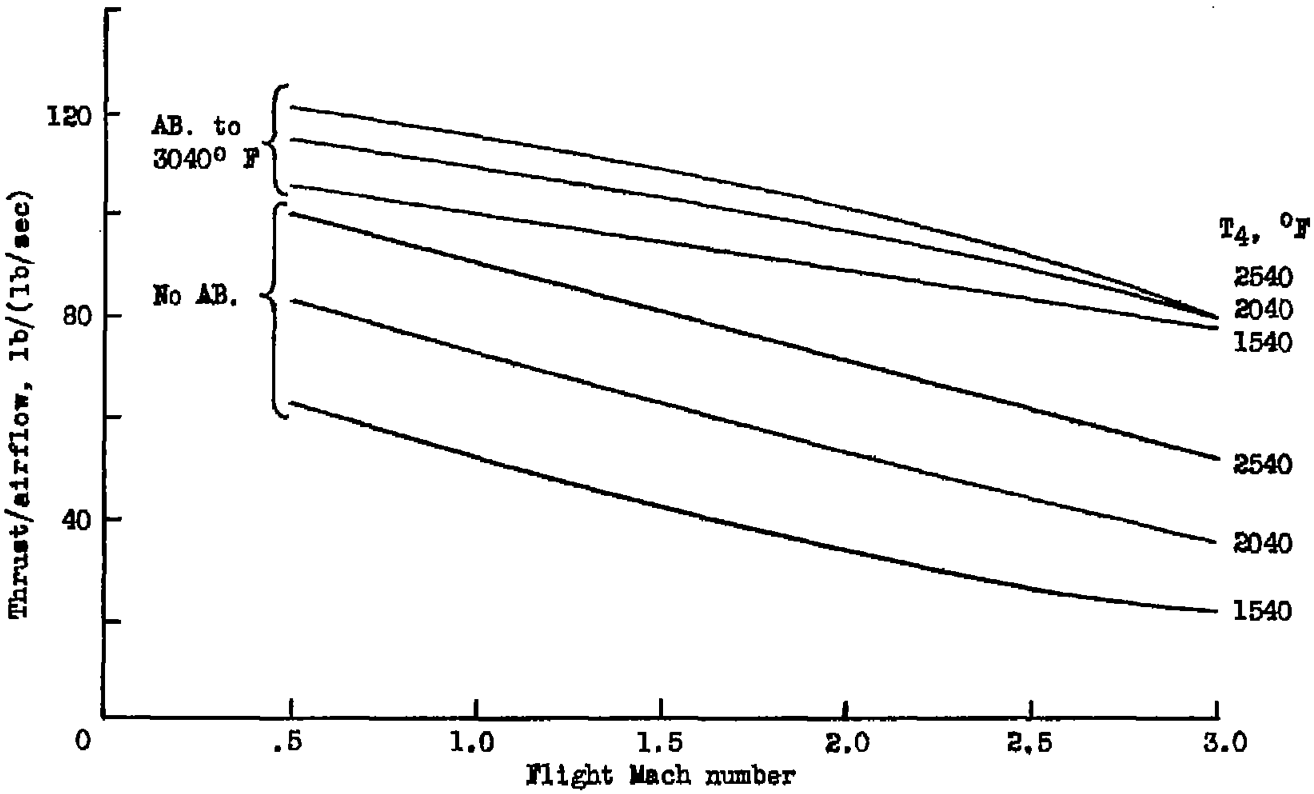
(1) All radius calculations are based in the Breguet range equation and as such give values somewhat different than those shown in company performance reports or the USAF "Green Book." The numbers shown in the table provide comparative radius values.

(2) A typical mission is a total radius of 1640 n. miles of which 200 n.m. are dash at M=2.0. The remaining radius of 1440 n.m. is flown at M=.9.



(a) Overall engine efficiency.

Figure 1.- Effect of flight Mach number on engine performance with and without afterburners. Altitude 35,000 feet and above.



(b) Thrust per unit airflow.

Figure 1.- (Concluded)

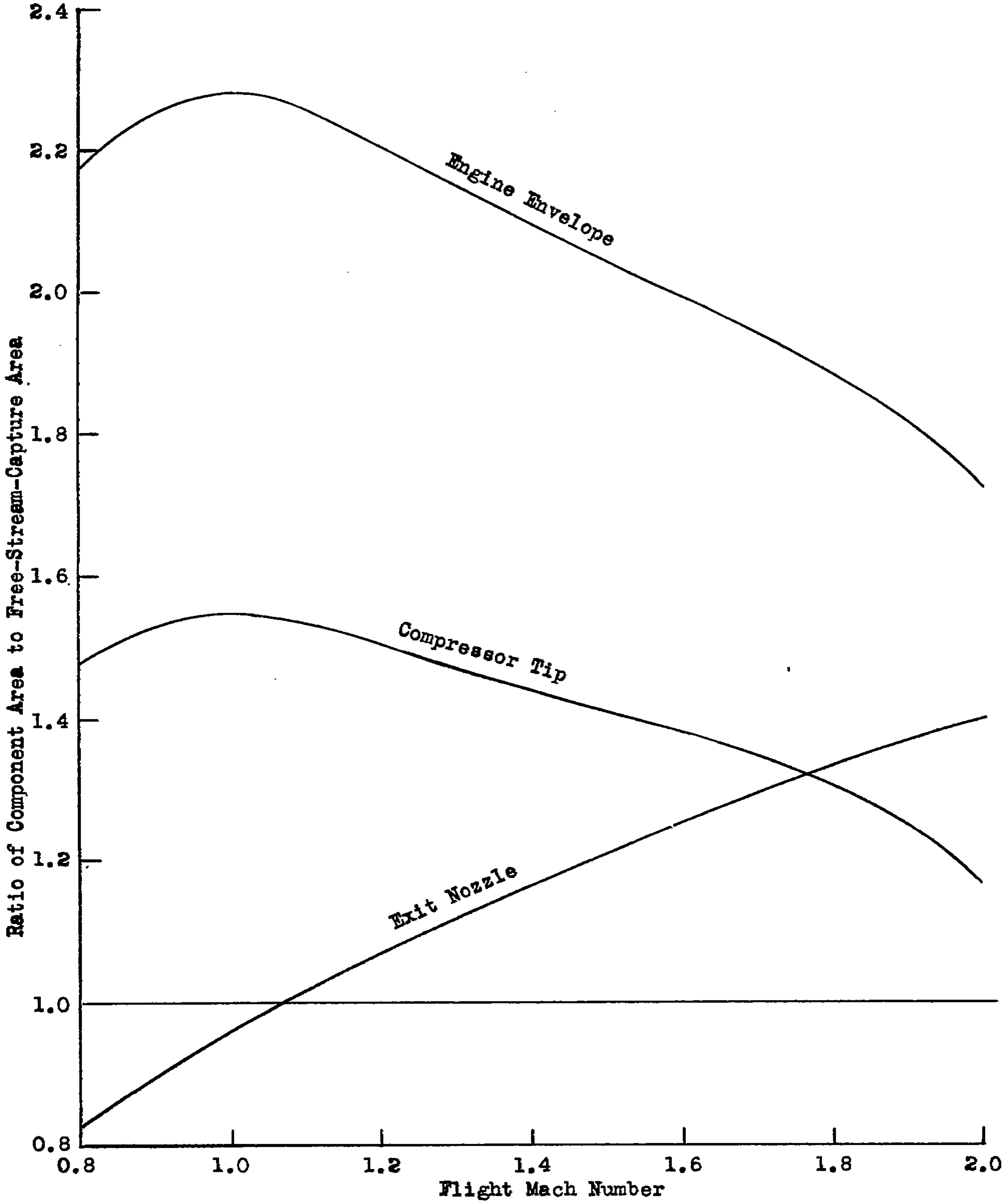


Figure 2 - Effect of Flight Mach Number on Engine Component Areas.

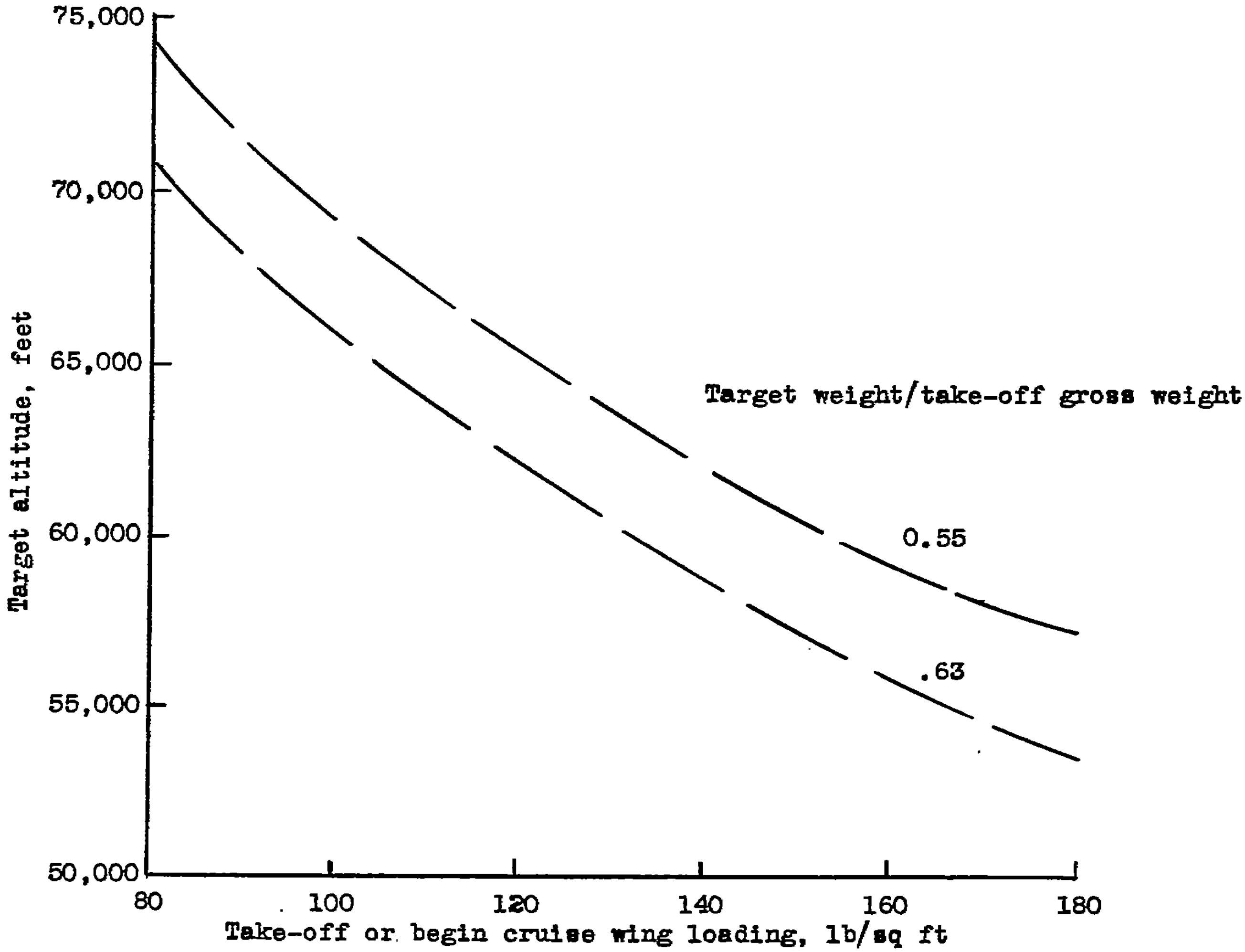


Figure 3.- Relationship of target altitude and take-off wing loading for bomber configurations. Lift coefficient, C_L, 0.2; flight Mach number, 2.0.

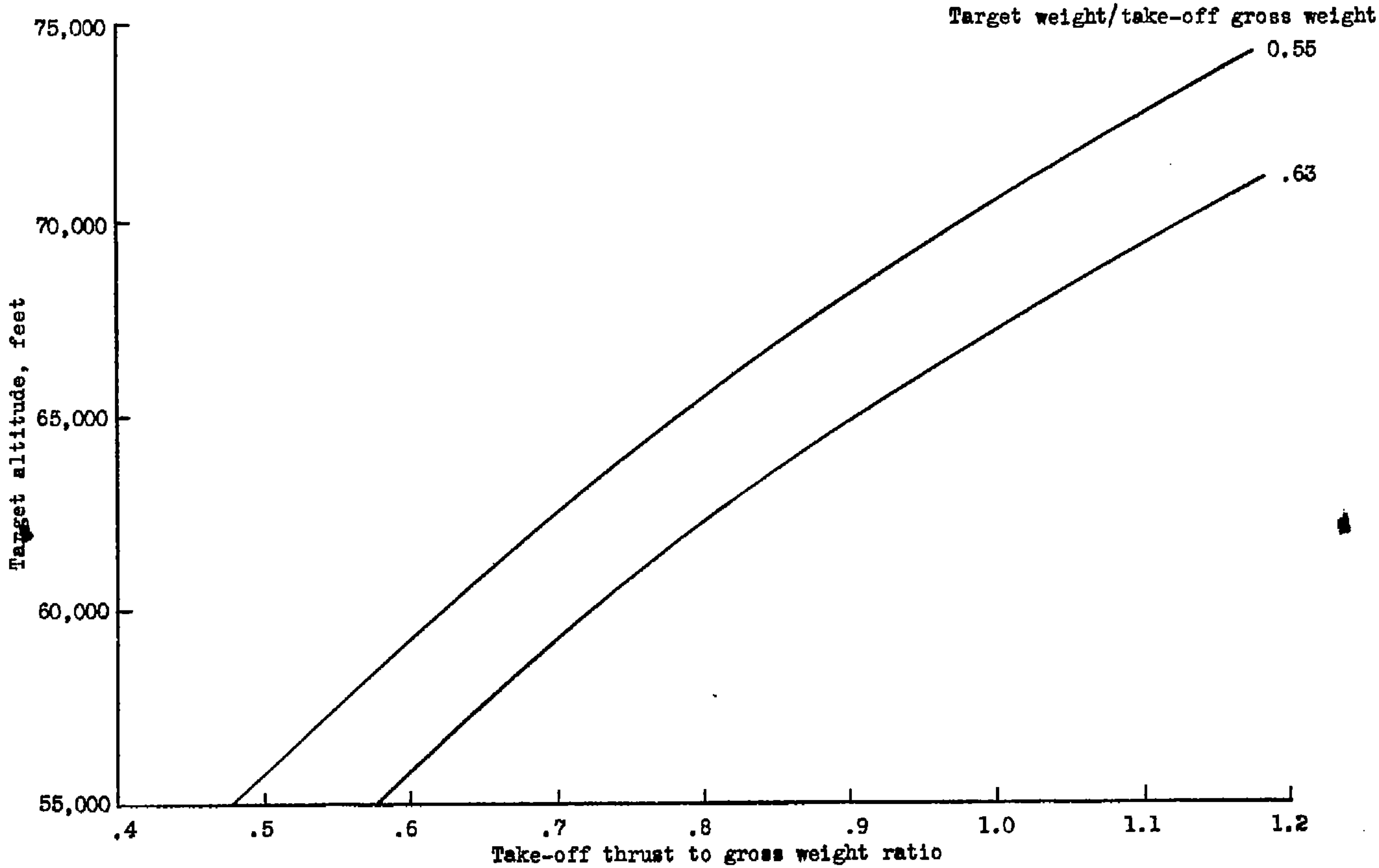


Figure 4.- Relationship between target altitude and take-off-thrust-to-gross-weight ratio for a representative non-afterburning engine. Flight Mach number, 2.0; lift-drag ratio, L/D, 5.0.

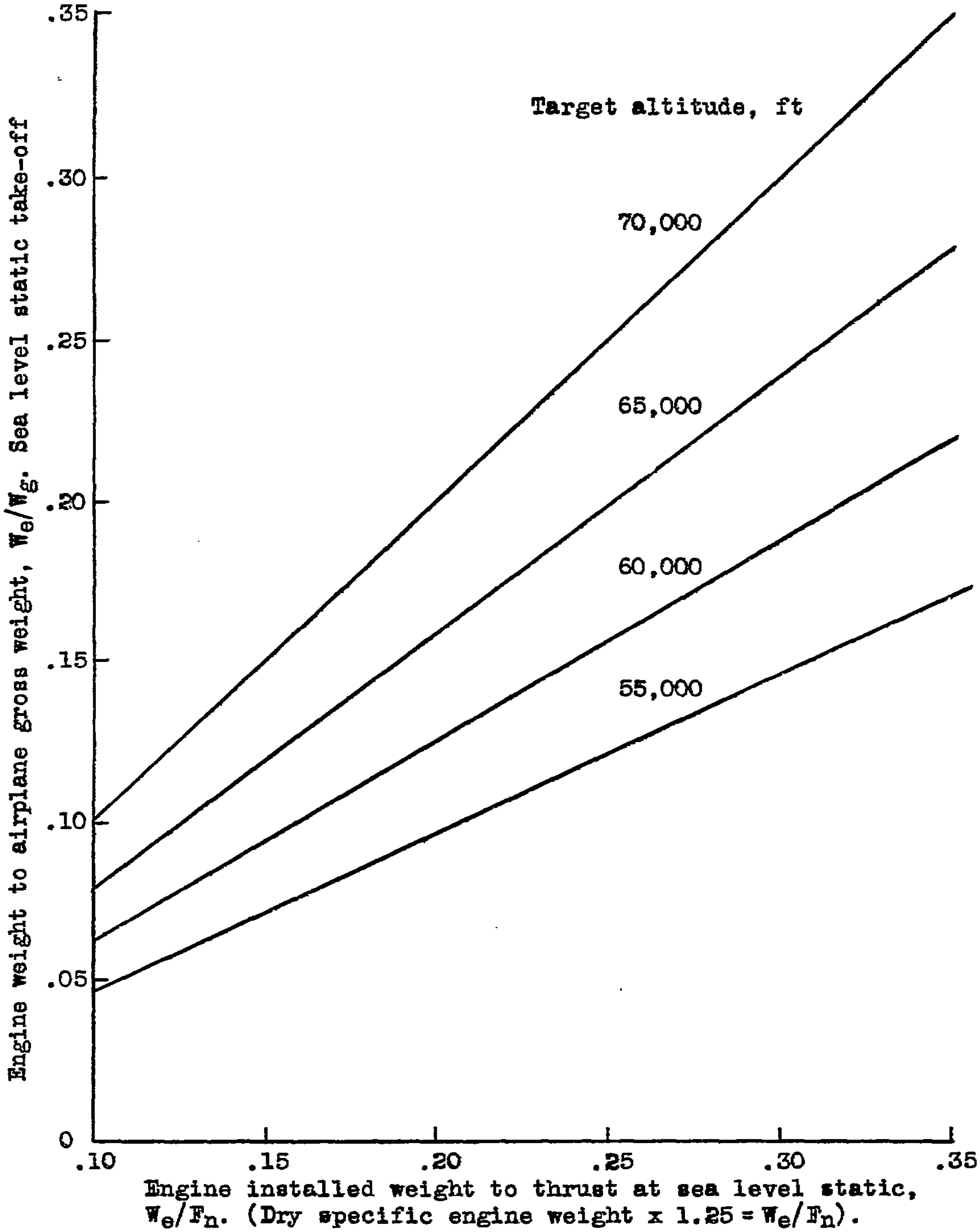


Figure 5.- Relationship of engine weight, airplane weight, and thrust for various target altitudes for non-afterburning T.J. engine. Flight Mach number, 2.0; Target weight = .55 take-off gross weight; L/D = 5.0.

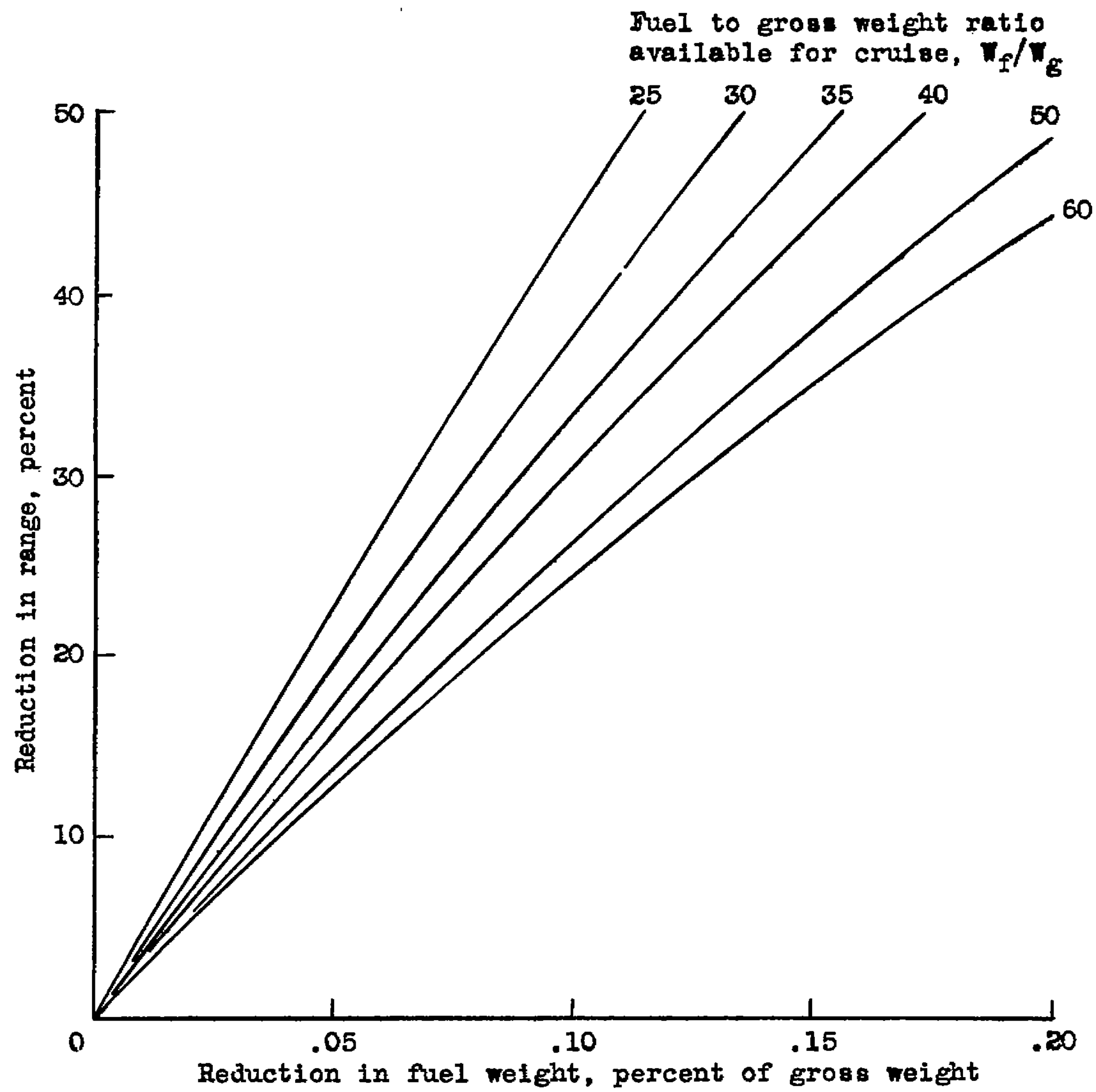


Figure 6.- Effect of fuel weight reduction on range.

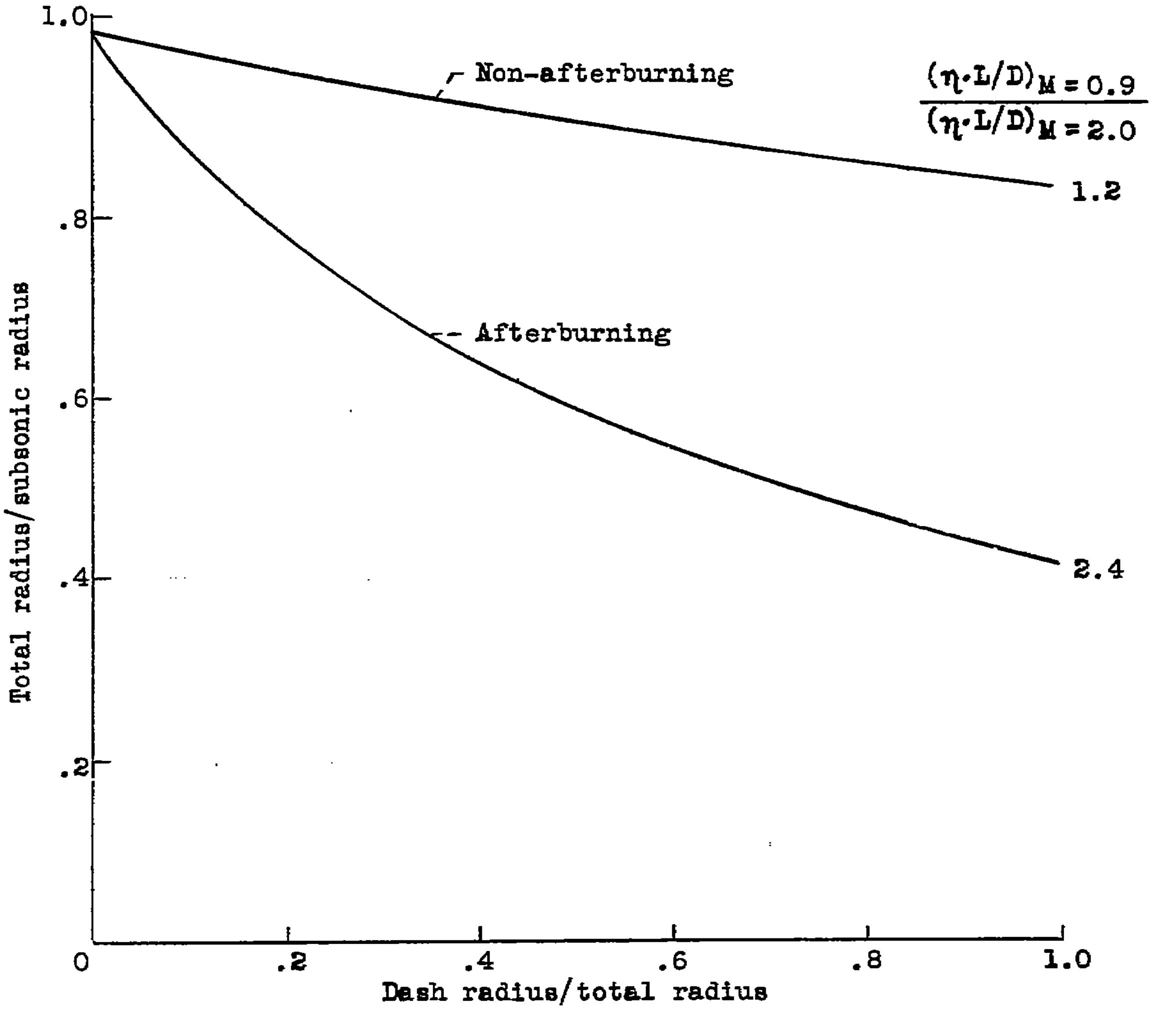


Figure 7.- Effect of dash radius on total radius.

Engines, Turbojet 3.1.3

Aircraft Performance 1.7.1.3

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Abstract

Airplane performance is analyzed to relate the effects of non-afterburning turbojet engine application to all supersonic flight and the resulting ranges of specific engine weight, overall engine efficiency, and airplane variables, i.e. L/D that make this engine application appear feasible.

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